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PRELIMINARY ANALYSIS OF MANNED MARS MISSION USING ELECTRIC PROPULSION SYSTEMS

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Introduction

In order to evaluate the role of electric propulsion systems for space missions, it will probably be necessary to determine their performance capabilities for many space missions and then compare these results with similar analyses for competing systems. Of the various missions that may be considered, it would seem reasonable to assume that the higher energy missions would favor electric propulsion systems with their high specific impulses. Unfortunately, one of the most interesting planets, Mars, has almost the lowest energy requirement of all the planetary missions. For this reason, it would not be surprising if electric propulsion systems did not show outstanding advantages for this particular mission. On the other hand, there are many other factors, in addition to energy levels and specific impulses, that must be evaluated before such a conclusion can be drawn.

Similar to other analyses of this type,[1] it is necessary to consider a number of different mission profile variations. For electric propulsion systems, few such extensive analyses have been made because of the difficult trajectory calculations, usually requiring the use of the calculus of variations and numerical integration. Recent work by Zola[2] has produced a method of making reasonable approximations in the trajectory area with 20:1 increases in computing speed. This advance allowed a relatively broad preliminary investigation to be made in the hope that it will point out new and interesting areas for future study.

Initial gross weight in Earth orbit required for a seven man crew has been chosen as the criterion to be minimized. These weight estimates will then be presented as functions of mission time for a 40-day exploration time at Mars. No attempt will be made to calculate the reliability associated with each mission, but some effort will be made to point out various areas of risk. Also, no attempt has been made to specify a departure date. Instead, Earth and Mars are assumed to be in circular orbits about the sun, resulting in fuel consumption that is intermediate relative to the best and worst launch dates.

Vehicle Design

A conceptual vehicle layout is shown in figure 1 where the usual attempt is made to keep the crew cabin at a distance from the reactor. Preliminary estimates of the required separation distances indicate that about 300 ft will be sufficient to bring the radiation hazard from the operating reactor down to a negligible level.

Figure 2 gives a more detailed view of the crew cabin design showing how the total volume of 5600 cu ft (800 cu ft/man) is divided between the living quarters and the heavily shielded solar flare shelter of 450 cu ft. In this particular design, the cylindrical section protruding from the floor may be moved up to the ceiling of the living quarters forming an added 550 cu ft of shielded volume for the long duration spirals through the Van Allen belts. A weight breakdown of a typical crew cabin design is given in Table I. As indicated, this breakdown includes estimates of the life-support requirement but does not include the radiation shielding, which is supplied mainly by onboard propellants as discussed later.

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Mars Payload

In the upper part of the cabin (fig. 2) there is a storage area provided for the Mars landing and exploration system and the Earth reentry vehicle, if needed. Weight estimates of the Mars landing system are given in Table II. All the vehicles shown start in a high circular orbit about Mars (27 Mars radii) and all except the tanker vehicle are sent directly to the surface by using a combination of atmospheric and chemical-rocket braking. The tanker is sent to a low circular orbit where it is joined later by the manned landing vehicles for refueling prior to the transfer back to the main spacecraft orbit. This saves the propellant that would otherwise be needed to transport the tanker load from the surface to the low orbit but requires that the crew perform an extra rendezvous with the tanker in the low orbit.

Shielding

As pointed out above, use has been made of propellants as a radiation shield. The propellant for the electric rocket is assumed to be mercury. In some cases, chemical propulsion is used just prior to atmospheric braking in order to limit entry speed. In this case, the propellant is assumed to be B_2H_6 and OF_2 and to have the shielding properties of water. Finally, in those cases where a solid shield is used, it is assumed to be polyethylene and also to have the shielding properties of water.

The dose accumulated from the Van Allen Belts was determined by using references 3 to 5 for the intensity, spectral data, and calculation methods, respectively. The number of solar flares encountered was determined by the statistical method of reference 1 by assuming one large flare every 4 years and one small flare each year. This gave the frequencies shown in Table III. The spectral and intensity data for the large type of flare were taken from reference 6, and the small flare was assumed to be half as intense as the large flare. In placing the flares, it has been pessimistically assumed that both a large and a small flare occur at the mission perihelion and that the intensity varies inversely with the square of the radius. Again the dose was computed by using the method of reference 5, which includes the effects of secondary neutron and proton production. Finally, an unshieldable input of 0.65 rem/week was assumed to result from cosmic rays.[7]

In all cases the crew was limited to a maximum instantaneous dose of 100 rem; however, it has also been assumed that some recovery occurs[8] from the ef-

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fects of a given dose.

Thrustor Performance and Weight Estimates

For most of this paper, the thrustors are assumed to be of the constant-thrust and specific-impulse type having the performance shown in figure 3. These curves are based on recent experimental data[9] for the electron-bombardment thrustor but have been modified somewhat to allow for some change in the state of the art. Figure 3(a) shows the losses (discharge only) as a function of that fraction of the propellant which is ionized. By optimizing this parameter, the efficiency curve shown in figure 3(b) is determined.

A simple estimate of thrustor weight is made by assuming a constant density of 300 kg/sq m of exit beam area. The exit area is then estimated by using the charge-exchange-current-density limitations[10] imposed to provide adequate grid life.

Trajectory Methods

As indicated previously, both Mars and Earth are assumed to be in circular, coplanar orbits about the sun. Furthermore, the total trajectory is treated as a series of two-body segments or phases. For the planetocentric phases, an approximation technique similar to that used in reference 11 has been employed. For the heliocentric phases, the method of reference 2 is used in all constant-thrust cases. This technique is essentially a method of correcting the ΔV of an easily obtained high-thrust solution for changes in thrust and specific impulses. (The high-thrust solution is computed by assuming that impulses occur at the start and/or end of the trajectory only.) In this work, both perihelion radius and entry velocities are needed in most cases. These were taken directly from the reference high-thrust solution and are not of the same quality as the corrected ΔV's. Some limited spot checking has been done which indicates that the radius and entry velocity values are always lower than those for true low-thrust calculations. This will tend to give somewhat higher required shield weights and slightly larger propellant loads required for braking prior to atmospheric entry, two trends which may tend to compensate for each other.

Nominal Mission Profile

After some preliminary investigations, the standard or nominal mission profile shown in figure 4 has been selected. Here, the mission begins in a polar circular orbit at 1.10 Earth radii and is followed by a spiral escape maneuver and a constant thrust (with intermediate coasting) transfer to the vicinity of Mars. The vehicle then spirals down to 27 Mars radii from which the landing is performed. After 40 days have elapsed, a similar transfer is made back to Earth where the vehicle spirals into a terminal circular orbit at 3 Earth radii, just beyond the inner Van Allen belt.

As shown in figure 4, the mission perihelion occurs during the return trip where the propellant supply on board is low. Since a solar flare is assumed to occur at perihelion, the crew has less protection from the propellant than if the perihelion occurred during the outbound trip. For variable thrust systems, it has been found[12] that the placement of the perihelion has no effect on propellant consumed. For the constant-thrust system used herein, there is appreciable propellant saving associated with return trip perihelion placement. This more than compensates for the loss of propellant protection associated with placement on the outbound journey. Consequently, all constant-thrust profiles will use this feature in the remainder of this paper.

The gross weights required for the nominal profile described are shown in figure 5 as a function of mission time. From this figure, it can be seen that gross weights of about a million pounds are possible, but only at mission times beyond about 600 days. Shorter trip times are possible, but the associated gross weights increase rapidly.

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Unmanned Belt Traversal

Early in this study, it became apparent that a large shield-weight saving could be had at the longer mission times by avoiding a manned traversal of the inner Van Allen radiation belt. This may be done by assuming that the crew remains on Earth while the main vehicle traverses the belts unmanned. Later, the crew rendezvouses with the main craft beyond the inner belt via a high-thrust transfer. The weight of the additional vehicle requires is negligible compared with the main vehicle, so that the major problem with this scheme is the added operational complexity; however, as indicated in figure 6, the weight saving is so large that it seems well worth the added initial rendezvous. Also shown in this figure is the time saved by not having men on board during the belt traversal.

For all of the following mission profiles, this method of avoiding the inner radiation belt will be used and the time saved accounted for.

Effect of Thrustor Performance

Two features of the assumed thrustor performance tend to detract from mission capabilities. First, it may be possible at some future time to operate much closer to 100 percent efficiency. Secondly, it may also become possible to gain further propellant saving by operation at variable thrust and constant power.[12] The relative importance of these separate effects is shown in figure 7, where it should be noted that most of the weight reductions result from elimination of the thrustor inefficiency and not from the variable-thrust feature. It appears, therefore, that significant future gains are possible through thrustor efficiency improvements, particularly in terms of lowering the minimum mission times possible with a given gross weight.

Atmospheric Braking

A recognized method for reducing mission ΔV requirements is to introduce atmospheric braking at return to Earth. Although commonly applied to chemical- and nuclear-rocket systems, it may also be used with electric systems in a variety of ways as indicated in figure 8. In addition to the case with no atmospheric braking (taken from fig. 6 or 7) the entry velocities of 37,000 and 52,000 ft/sec are

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also shown. These velocities have been maintained by using either electric or chemical (B_2H_6 plus OF_2 with I = 430 sec) deceleration prior to entry. In the case where electric deceleration is assumed, the increase in entry speed reduces the gross weight and shifts the curves to the left. In the case of chemical deceleration and atmospheric braking, far too much propellant is needed for the lower entry speed (not shown) to be of any service. For the higher speed, however, the chemical braking scheme is somewhat superior to the all-electric scheme. It would appear, then, that there is some entry speed for which the two methods are equivalent.

Although entry speeds higher than 52,000 ft/sec are considered feasible,[13] the limit chosen here is believed to be a reasonable compromise between increased performance and risk, and will be used, with chemical braking, for the rest of this analysis.

Two-Phase Missions

In the previous figures, it was noted that short-duration missions required high gross weights because of the high propellant requirement, which suggests sending ahead, on a separate trip, material not needed by the crew on the outbound trip. By this procedure, part of the nominal payload would be sent via a more efficient, long-duration trajectory resulting in less total propellant requirement. This method should, therefore, be most helpful for the short-duration trips. Two cases of this type are considered in figure 9. In the first case, only the Mars payload and landing system is sent ahead on a 350-day transfer. The weight of the first-phase vehicle is computed by using the same specific powerplant weight and best travel angle possible for the 350-day duration. The weights shown on the ordinate are the sum of the weights of both vehicles. In the second and best case, the propellant for the return trip is added to the previous first-phase payload.

The major effect of this profile is to reduce gross weights mainly for the shorter missions as predicted above. These gains are rather small, however, and may not warrant the added risk and complexity required. This is particularly true in the case where the return propellant is sent ahead.

Effect of Specific Powerplant Mass

In order to place electric propulsion systems in proper perspective, consideration must finally be given to one of its major unknowns - specific powerplant mass. The nominal value of 7 kg/kw used here is based on current estimates[14] that range between 4 and 10 kg/kw; however, it must be recalled that no such system has ever been built.

Figure 10 shows how the relation of gross weight to mission time is effected by specific powerplant mass. At the very long mission times, around 600 days, the impact of this parameter is relatively small but becomes more important as the mission time is reduced. The primary effect, however, is to reduce the apparent minimum mission time.

Comparison with Other Systems

One method of avoiding the long-duration spiral and Van Allen belt hazard is to accomplish the Earth escape phase with a high-thrust stage added to the basic electric system. Use of a nuclear rocket for this purpose corresponds to the combined high- and low-thrust system studied by Levoy[15] and Edelbaum.[16] For this type of vehicle, the optimum amount of high-thrust assist is determined on the basis of minimum gross weight (for both systems added together) as shown in figure 11 for the mission time of 400 days. In most cases this boost is sufficient to intercept Mars without any added propulsion. If the remaining propulsive phases (Mars arrival and departure) are supplied by the electric rocket, the weights shown in figure 12 result. Also shown here are the all-electric system discussed previously and an all-nuclear system, all for the same type of mission profile with $\alpha = 7$ kg/kw.

A comparison of the three systems shows that the combined system gives lower weights over the entire range of mission times considered, with a much larger advantage at the shorter mission times. Beyond about 450 days, however, there does not appear to be a sufficient weight saving to warrant the complication of an added nuclear stage. Below 450 days, the all-nuclear system surpasses the electric system, which, however, occurs in an area of great superiority for the combined system over both competing systems.

In addition to the weight saving shown for the combined system, there are also the associated reductions in power requirements shown in figure 13. A typical case (T_M = 550 days) shows a reduction from 5 to 1.8 Mw with an even larger saving possible at the shorter mission times.

Concluding Remarks

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This paper has estimated initial gross weights for a number of mission profiles for a seven man Mars mission. A nominal profile, with four spiral-type propulsion phases, can achieve weight as low as a million pounds if the mission time is allowed to increase to 650 days. Use of an unmanned belt traversal can reduce these weights to about 500,000 lb while saving about 40 days on the mission time. The introduction of atmospheric braking at Earth return can give further reductions to about 300,000 lb at 600 days while also reducing minimum mission time from 500 to 400 days. Finally, further but smaller reductions are possible by using two-phase profiles; however, these do not appear worth the concomitant risk and complexity.

Comparisons with the all-nuclear rocket and combined systems show that the combined system is superior over a wide range of mission times at a specific powerplant mass of 7 kg/kw. The electric system becomes equivalent to the combined system at the longer mission times and is superior to the nuclear-rocket system. For the shorter mission times, however, the combined system is far superior to the others.

Although this study indicates that the combined nuclear- and electric-rocket system is superior for this mission, it may also be said that the all-electric system is competitive, particularly at the longer mission times. Consequently, if electric systems are definitely competitive for the Mars mission, it is even more certain that they will be superior for more difficult ventures.

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Table I. - Crew Cabin Weight Estimates

(No Radiation Shielding Included)

Description	Weight, lb
Structure, meteorite protection, and thermal control	23,000
Furnishings	2,000
Centrifuge, 1/3 g	1,000
Repair, medical, and recreational facilities	2,000
Fixed life-support weight[a]	3,500
Contingency	500
Total	32,000

[a]The time dependent part of the life support system weight has been assumed to increase at a rate of 45.5 lb/day for the seven man crew.

Table II. - Mars Payload Weight Estimates

Description	Weight, lb
Two manned landing vehicles	53,000
Two equipment landing vehicles	14,600
Tanker	9,200
Scientific equipment and probes	6,500
Miscellaneous	700
Total	84,000

Table III. - Assumed Frequency of Solar Flares (1.0 Percent Risk of Exceeding Each Type)

Exposure time, days	Small type, 1/yr	Large type, 1/4 yr
160 to 210	2	1
210 to 300	2	2
300 to 462	3	2
462 to 630	4	2
630 to 654	4	3
654 to 800	5	3

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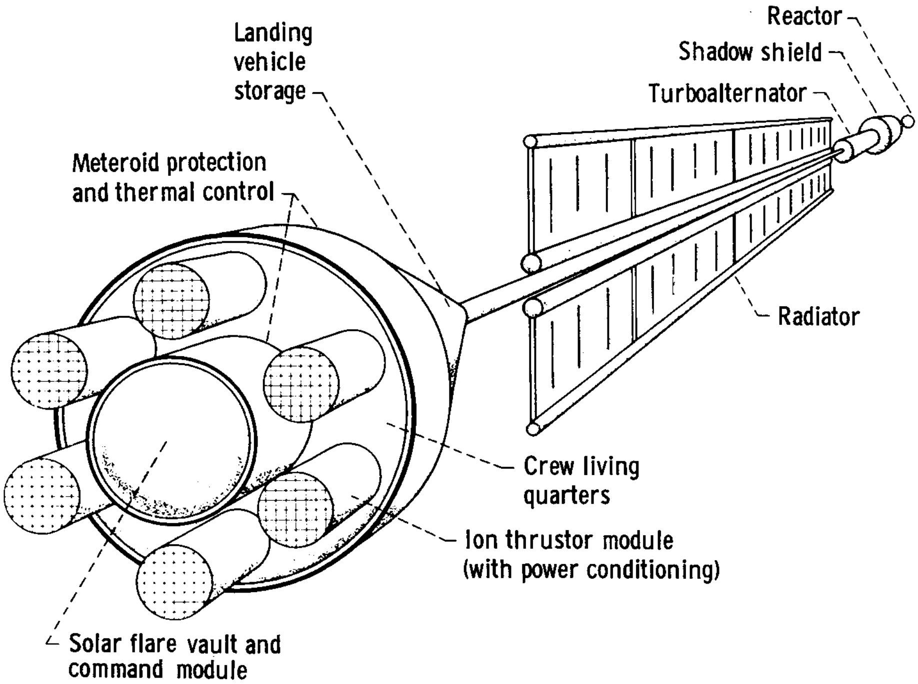


Figure 1. - Conceptual design of electric propulsion system for manned Mars mission.

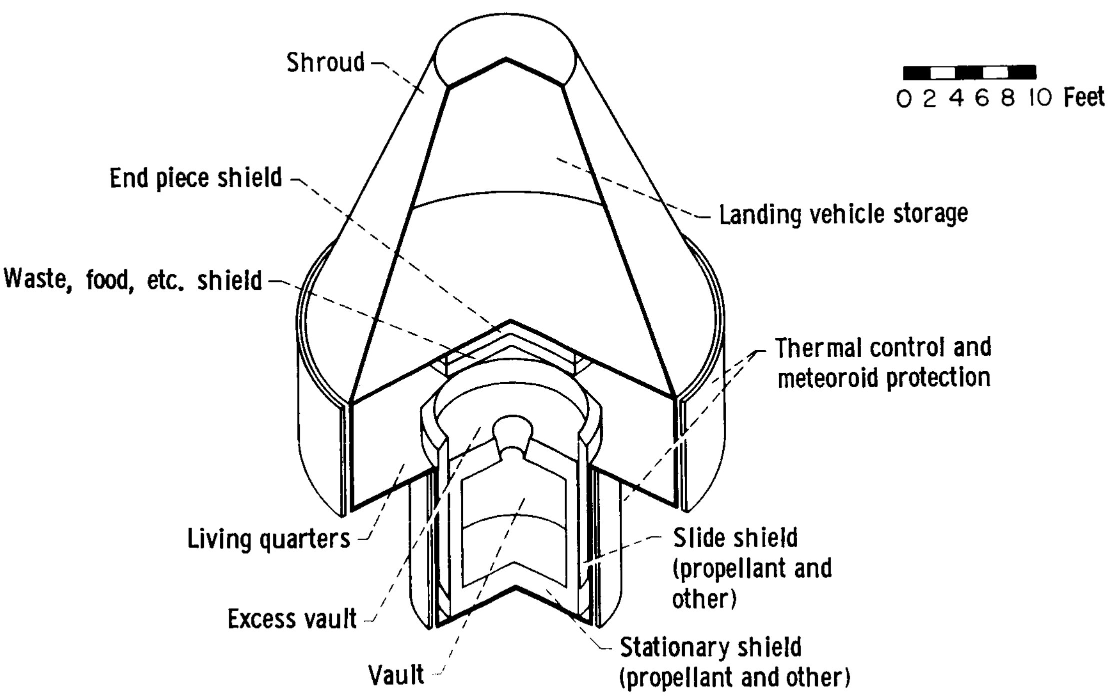
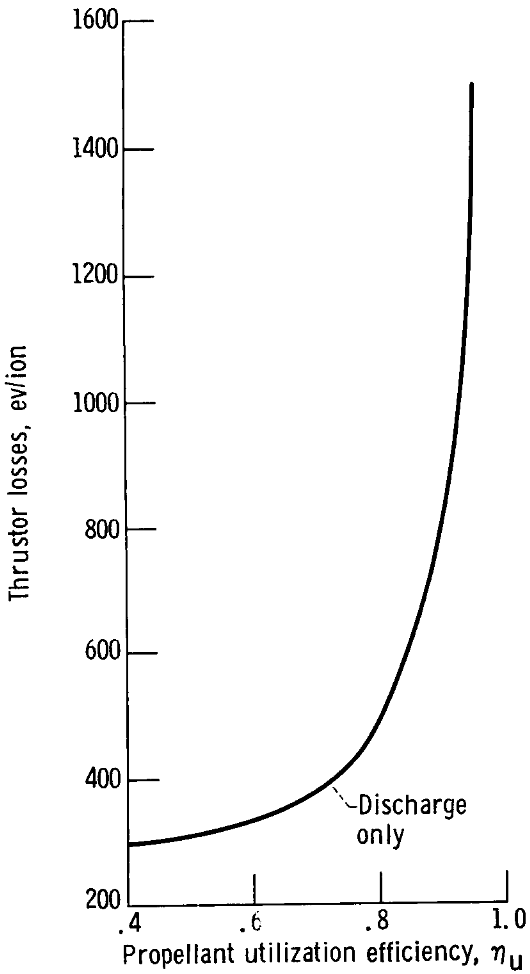
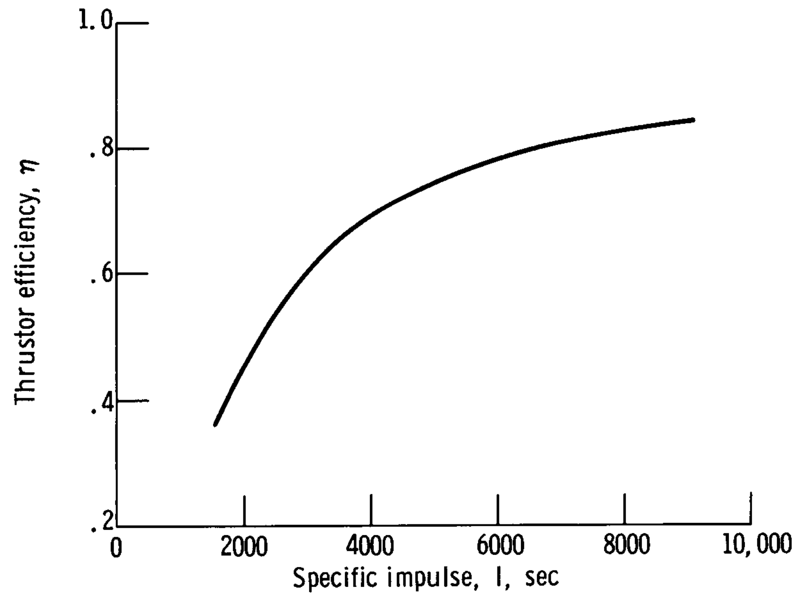


Figure 2. - Typical crew cabin configuration for seven man Mars mission.



(a) Thrustor losses.

Figure 3. - Thrustor performance characteristics for mercury electron-bombardment thrustor.



(b) Maximum thrustor efficiency.

Figure 3. - Concluded. Thrustor performance characteristics for mercury electron-bombardment thrustor.

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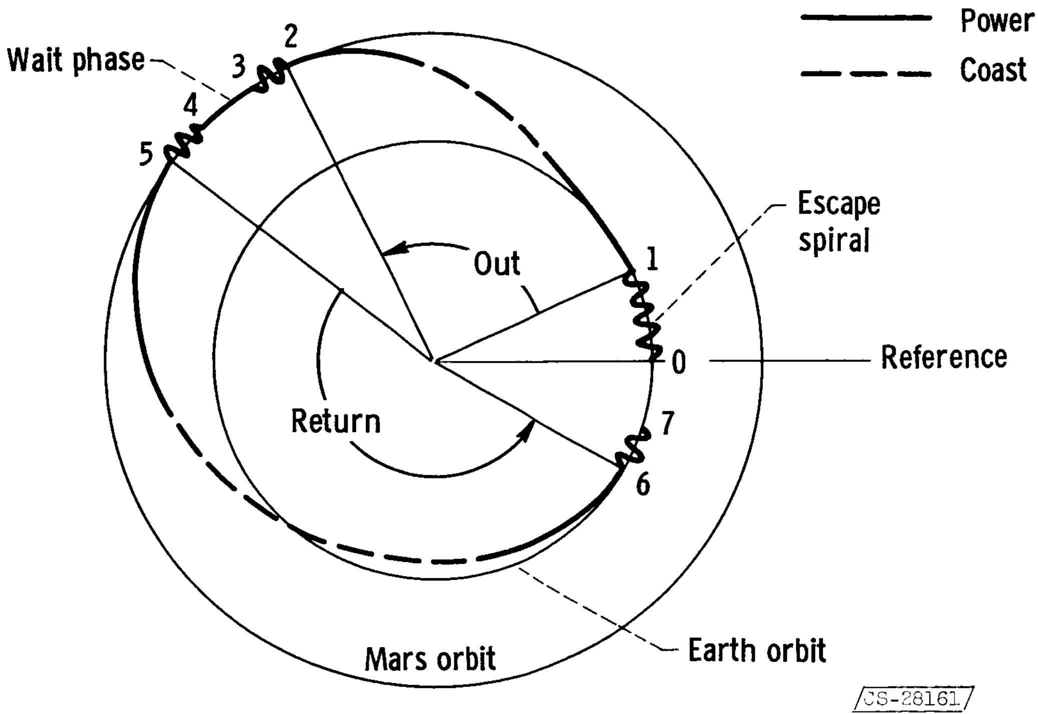

Figure 4. - Nominal mission profile for manned Mars mission using electric propulsion.

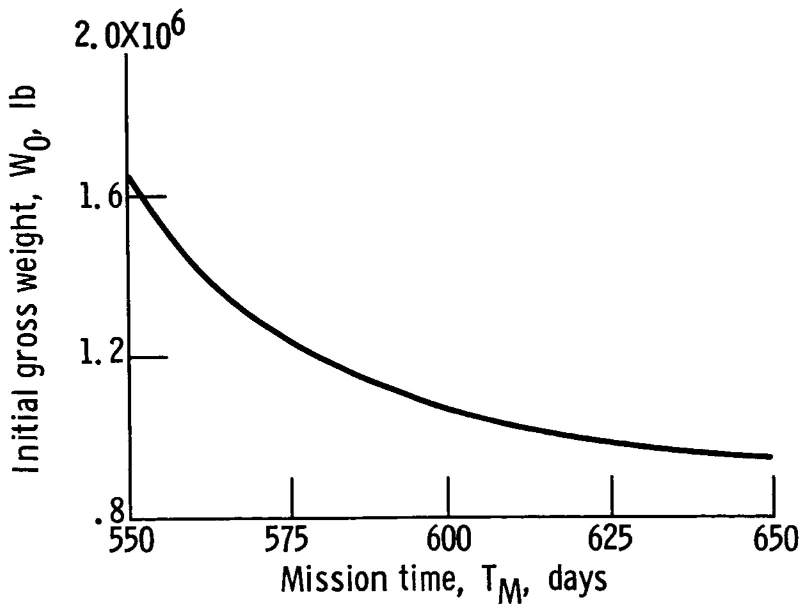

Figure 5. - Performance using nominal mission profile for seven man Mars mission. Wait time, 40 days; specific powerplant mass, 7 kg/kw.

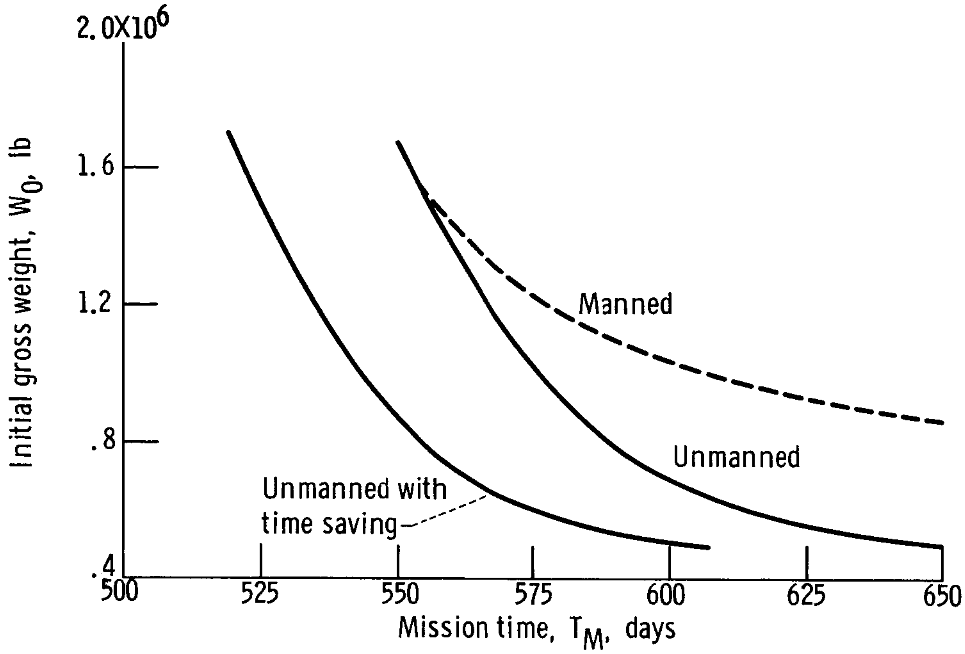

Figure 6. - Comparison between manned and unmanned belt traversal for seven man Mars mission. Wait time, 40 days; specific powerplant mass, 7 kg/kw.

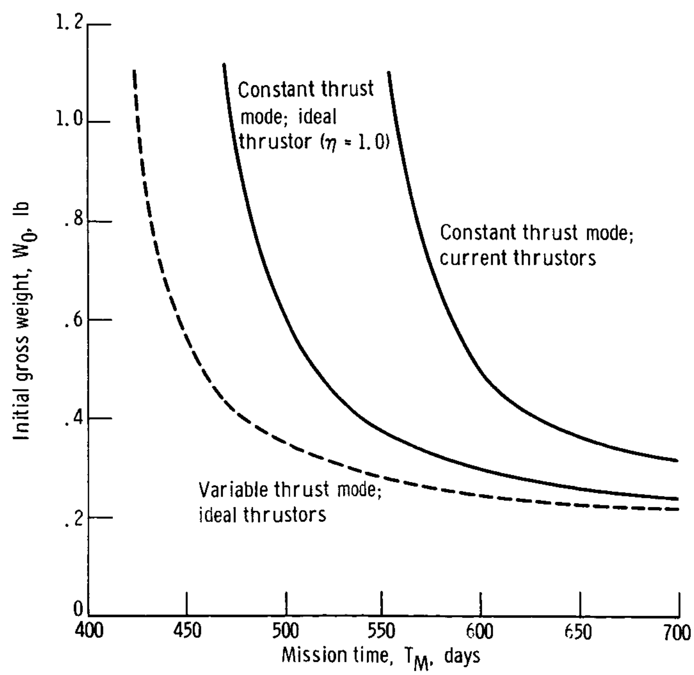

Figure 7. - Effect of thrustor type and efficiency for seven man Mars mission. Unmanned belt traversal. Wait time, 48 days; specific powerplant mass, 7 kg/kw.

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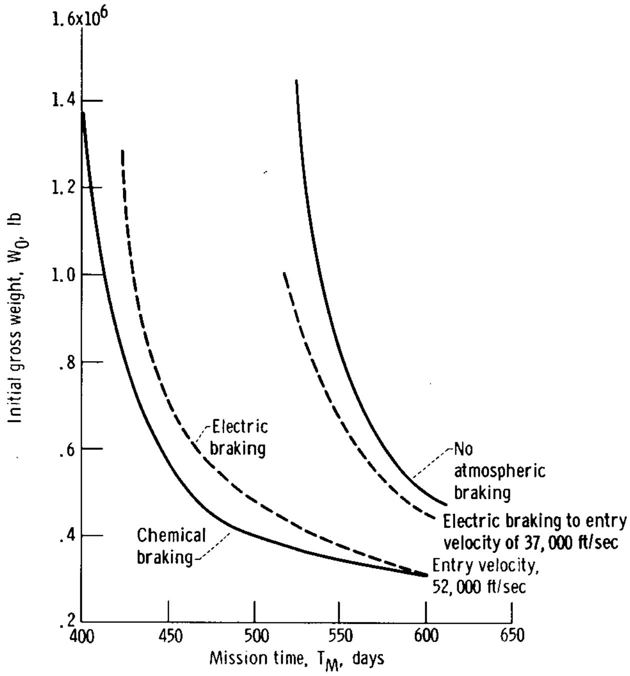

Figure 8. - Effect of atmospheric braking for seven man Mars mission. All-electric system; unmanned belt traversal. Wait time, 40 days; specific powerplant mass, 7 kg/kw.

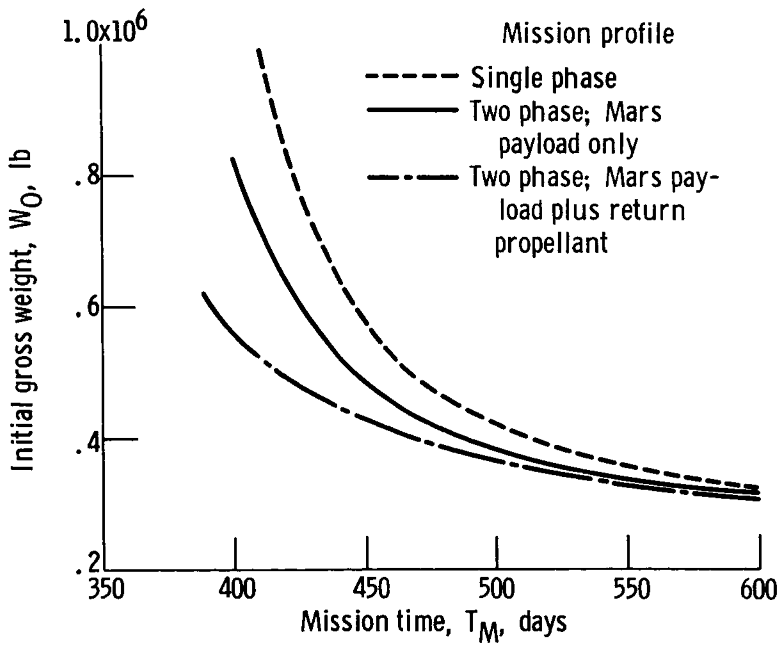

Figure 9. - Comparison of single- and two-phase mission profiles for seven man Mars mission. Unmanned belt traversal. Wait time, 40 days; specific powerplant mass, 7 kg/kw; entry velocity, 52,000 ft/sec.

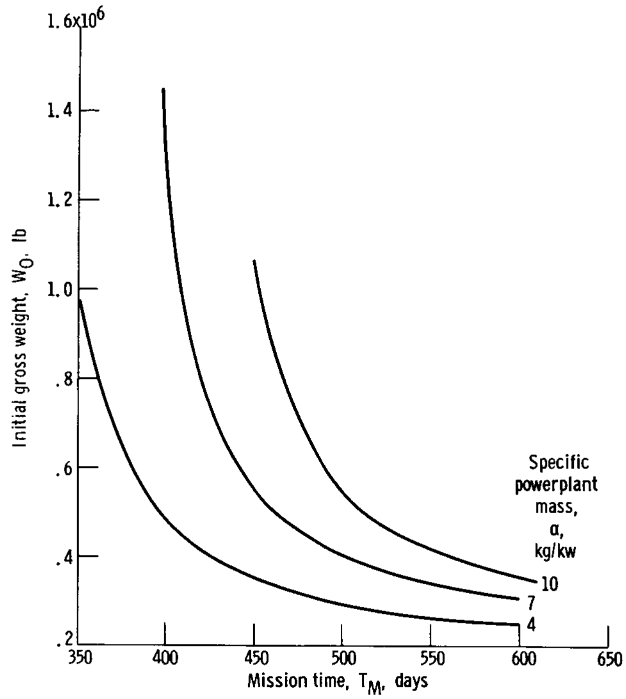

Figure 10. - Effect of specific powerplant mass for seven man Mars mission. Unmanned belt traversal; chemical braking to 52,000 ft/sec. Wait time, 40 days.

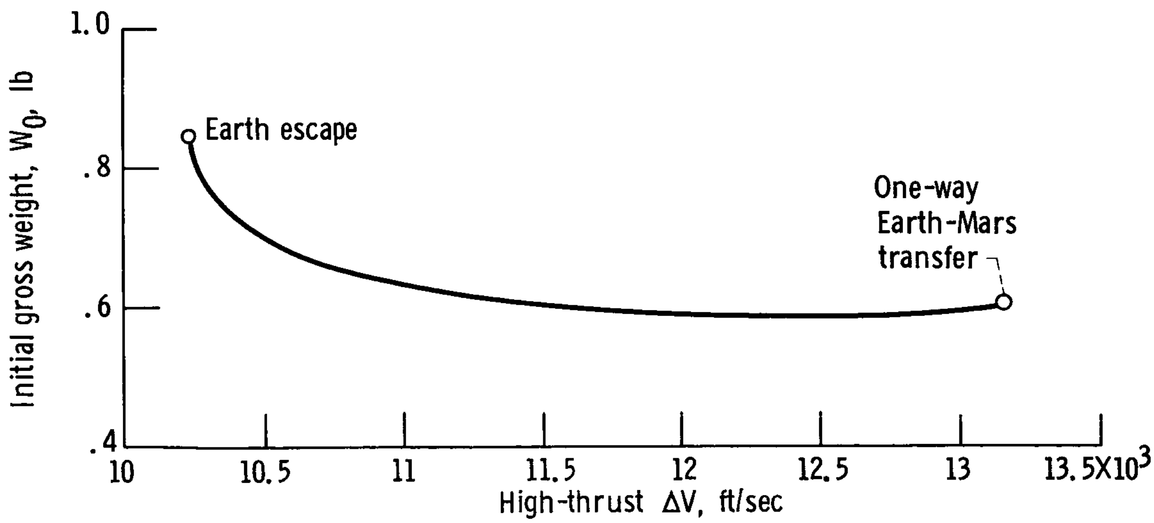

Figure 11. - Effect of high-thrust propulsion on combined system gross weight for seven man Mars mission. Wait time, 40 days; specific powerplant mass, 7 kg/kw; entry velocity; 52,000 ft/sec; high-thrust specific impulse, 850 sec; mission time, 400 days.

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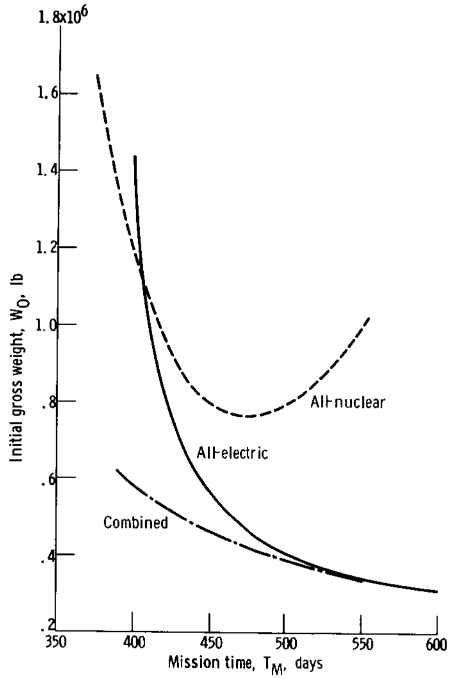

Figure 12. - Comparison of all-electric, all-nuclear, and combined rocket systems for seven man Mars mission. Wait time, 40 days; specific powerplant mass, 7 kg/kw; entry velocity, 52,000 ft/sec.

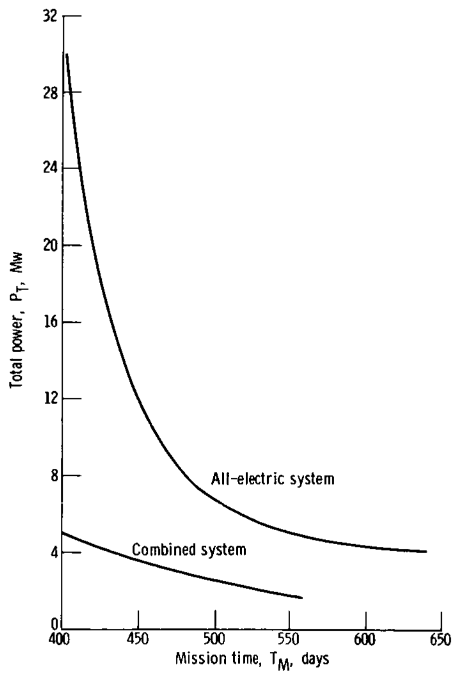

Figure 13. - Comparison of total power requirements for all-electric and combined systems for seven man Mars mission. Wait time, 40 days; specific powerplant mass, 7 kg/kw; entry velocity, 52,000 ft/sec; nuclear rocket specific impulse, 850 sec.